

ANALYSIS AND SIMULATION OF THE MISSION 3B RENDEZVOUS

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Important assistance was also rendered by Lt. Col. Karol J. Bobko of the Astronaut Office, and Maj. Byron T. Theurer of the Air Force Shuttle Office, who served as simulator test subjects.

Development of reference trajectory was a joint effort with Flight Analysis Section of MPAD. Their analyses are documented in a separate Internal Note.

LIST OF SYMBOLS

1. SUMMARY

Mission 3b consists of the retrieval of a large payload in near-Earth orbit, followed by a return to the launch site within one revolution. This mission is probably the most unique Earth orbital mission yet attempted in the space program; it compresses into one orbit several mission operations which in normal circumstances would be accomplished in several revolutions.

It was believed that analysis of this mission would reveal several drivers in crew related functional requirement areas. The expected result was that rendezvous related requirements would be discovered, and that other system requirements might be latent within this mission. Rendezvous sensor characteristics, mission design, display data, vehicle thrust to mass ratios, RCS propellant requirements, and IMU performance are areas potentially affected. The basic objective of the study was to reveal and quantize those unique drivers inherent in mission 3b.

A three-phase approach was outlined for the accomplishment of the study. First, a mission investigation was performed to establish basic trade-offs of thrust to mass (T/M) ratios, transfer time, crew workload, and other parameters of interest against the basic constraints imposed on mission 3b. After the selection of a basic trajectory, braking schedule, and crew loading parameters, a Monte Carlo analysis was performed utilizing expected IMU and rendezvous sensor performance. Finally, a real-time simulation was performed to verify analytical results, establish attitude propellant requirements, determine degradation due to loss of display information, and secure pilot comments.

Several conclusions have been reached:

a. Results have indicated a transfer interval of 72° of orbit travel meets most constraints.

b. A RCS T/M of $.3 \text{ fps}^2$ in X and Z and $.2 \text{ fps}^2$ in Y is the minimum requirement for this mission.

c. The loss of display information, specifically line-of-sight rate and range rate, significantly affects propellant consumption, arrival time, and crew workload.

d. IMU performance requirements typical of the assumed characteristics of the study should be adopted.

e. The rendezvous sensor should be capable of acquiring the target within a 30° half-angle cone.

f. RCS consumption is within current tank sizing constraints.

g. The allowable duty cycle is a function of display information available to the pilot.

h. Some unique attitude control modes may be required for this mission.

2.0 INTRODUCTION

2.1 Major Events

The once-around rendezvous mission (reference 1) is characterized by the following major events:

- a. Launch (WTR)
- b. Insertion
- c. External Tank Blow-down
- d. External Tank Jettison/Deorbit
- e. Rendezvous Sensor Acquisition
- f. Mid-course Correction (if required)
- g. Terminal Braking
- h. Payload Retrieval/Stowage
- i. Shuttle Deorbit
- j. Entry
- k. Landing (WTR)

2.2 Constraints

The constraints currently identified as governing the conduct of the mission are:

- a. External Tank Deorbit

As identified by reference 2, the external tank SRM deorbit ignition must take place no later than 240 seconds after insertion.

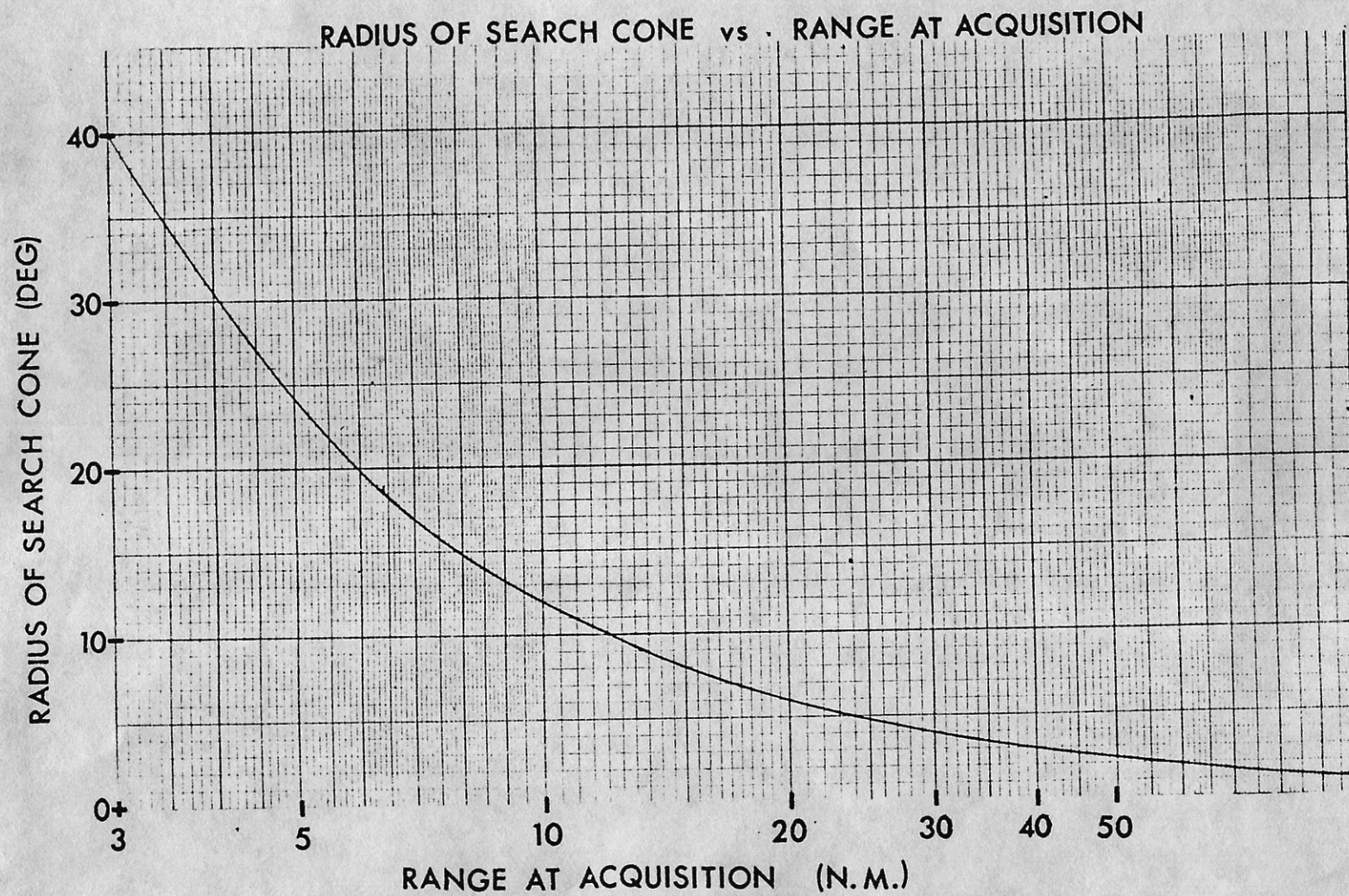
- b. Range at Insertion

(1) For a given time to intercept (relative to insertion), the insertion range must not be so large that the braking delta-V exceeds the RCS capability.

(2) The area of the target acquisition uncertainty cone must not exceed the capability of the rendezvous sensor to search and acquire before terminal braking. The size of this cone as a function of range is depicted in figure 2.2.1.

(3) Tank deorbit SRM ignition must not contaminate the target with exhaust material. This required a minimum range of 6 nautical miles at ignition as specified by SAMSO.

FIGURE 2.2.1



c. Insertion Dispersions

The principal source of position and velocity errors at insertion is IMU alignment errors at launch. Guidance errors are assumed to fall within this envelope.

d. Payload Visibility/Tracking

Mission 3B must be accomplished under all lighting conditions (SAMSO). Onboard tracking of the target is required immediately after tank jettison (reference 3).

e. Arrival Time at Intercept

The latest time of intercept is defined by the nominal time of Shuttle deorbit minus the time required to complete payload operations (reference 6). Intercept is considered to be achieved when distance to the target has decreased to 50 feet and relative motion has been nulled.

f. Duty Cycle

Duty cycle is defined as the fraction of time in the braking phase that the thrusters are turned on for translation. This percentage must not be so large that insufficient crew time is available for monitoring relative position and rate information.

2.3 Intercept Trajectory

For all parts of this study, initial conditions were chosen such as to result in an approach along a 135° elevation line during terminal braking (figure 2.3.1). This causes the terminal inertial line-of-sight rates to be near zero, permitting use of line-of-sight control (IOSC) braking. In this scheme, the pilot nulls apparent inertial motion normal to the line of sight, and controls the closure rate to be within a dead-band specified as a function of range. These control operations are normally initiated upon arrival at specific range gates.

It can be shown that the approach direction at intercept is a function only of the transfer interval, wt , and the elevation (E_{TPI}) at the start of terminal phase initiation. Also, having chosen wt and E_{TPI} , the closure rate at intercept is a linear function of the initial range.

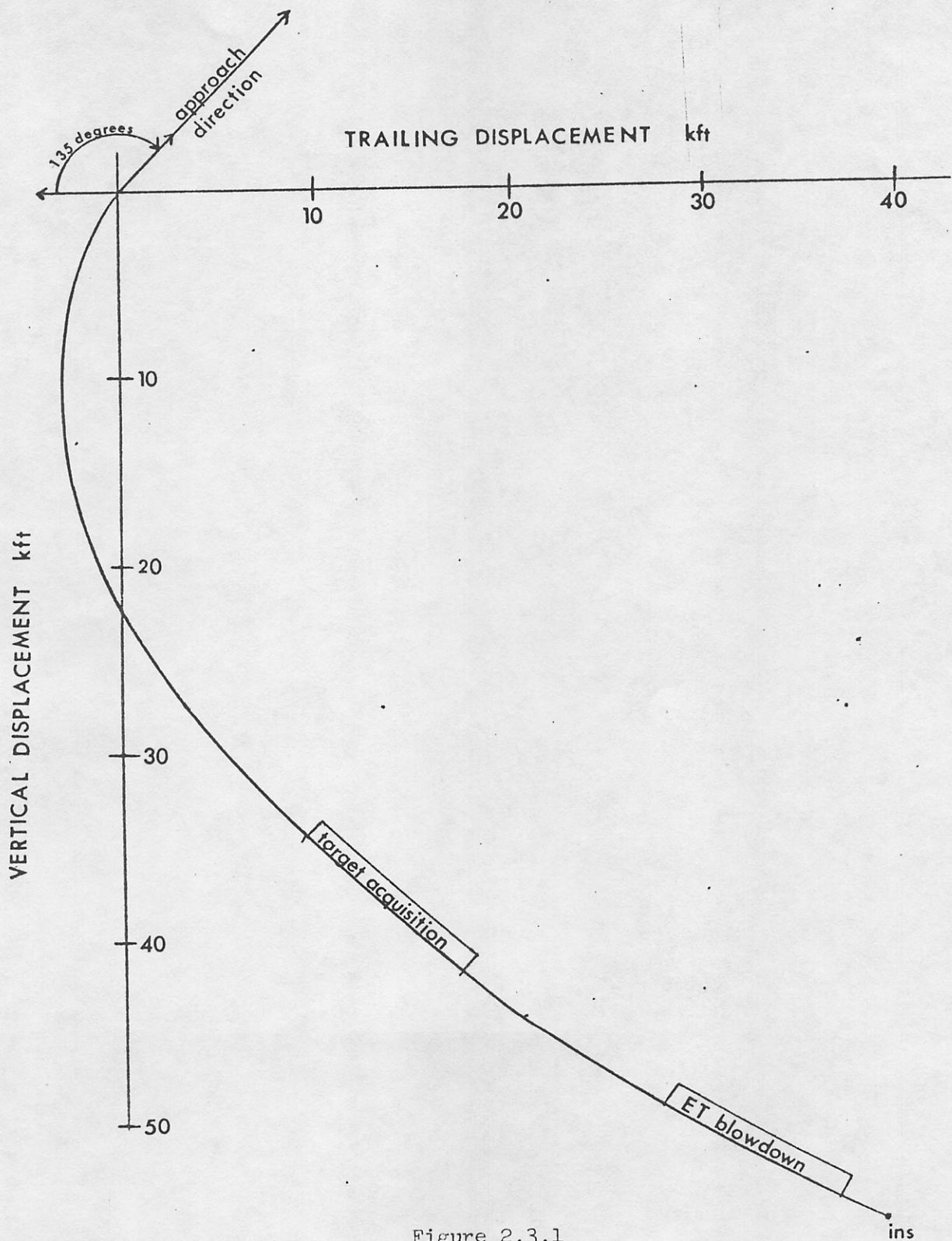


Figure 2.3.1
RENDEZVOUS GEOMETRY

Generally speaking, the longer the transfer interval, the lower the closure rate. Choice of transfer interval is limited by the time of arrival constraint on the high side, and an additional allowance must be made for lengthening of the transfer due to braking. For these studies, a design goal 25 minutes after insertion was selected for the 3- σ latest time of intercept.

All cases studied were common in the following respects:

- a. Insertion onto an intercept
- b. Time at intercept $\pm$ 25 min., 3- σ
- c. Approach along an elevation line of 135°

2.4 Initial Conditions

For this study, the target was assumed to be in a 100 n.m. circular orbit, with appropriate inclination. The covariance of state uncertainties is given in table 2.4.1, taken from reference 4. The various intercept cases were constructed by specifying initial elevation of the target and wt at Shuttle insertion. Table 2.4.2 presents the position, velocity, and correlated platform misalignment errors used for the Monte Carlo study (reference 5). Parametric and Monte Carlo studies, discussed in Sections 3.0 and 4.0, were used to establish values for initial elevation and wt which met the constraints.

2.5 Study Phase

After specification of constraints (Section 2.2) and choice of a general study approach (Section 2.3), a series of parametric studies were conducted to determine the impact of the constraints on the conduct of the mission. These studies resulted in a braking schedule and an envelope of transfer intervals and insertion ranges to be subjected to Monte Carlo analysis.

At the completion of the statistical investigation, a nominal initial condition for the spacecraft was chosen for the Shuttle Procedures Simulator (SPS). Also selected from the Monte Carlo runs were a group of dispersion cases.

R	T	C	DR	DT	DC
END OF TRACK +3 HOURS					
.15156E+06					
-.13264E+07	.13441E+08				
-.13225E+05	.15304E+06	.24803E+05			
.14617E+04	-.14838E+05	-.17233E+03	.16389E+02		
-.10256E+03	.84288E+03	.87145E+01	-.92948E+00	.73457E-01	
.36138E+01	-.77049E+02	.86978E-01	.83558E-01	-.77959E-03	.36467E-01

SQUARE ROOT OF DIAGONAL

.38931E+03	.36662E+04	.15749E+03	.40483E+01	.27103E+00	.19096E+00
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RSS OF POSITION AND VELOCITY

.3690173068E+04 .4061901100E+01

$\hat{\underline{R}}$ = Unit ($\underline{R}_T$)

$\hat{\underline{T}}$ = Unit ($\hat{\underline{C}} \times \hat{\underline{R}}$)

$\hat{\underline{C}}$ = Unit ($\underline{R}_T \times \underline{V}_T$)

Table 2.4.1
3b Target Covariance

Insertion

Optics azimuth alinement and accelerometer tilt leveling

TOTAL COVARIANCE MATRIX

U	V	W	U	V	W	O	O	U
.141+07	-.291+06	-.254+05	.879+04	-.168+04	-.146+03	.616+03	-.230+04	.364+05
	.866+05	-.991+05	-.181+04	.491+03	-.593+03	.407+03	-.420+03	-.774+04
		.204+07	-.108+03	-.362+03	.122+05	-.699+05	.151+05	-.175+03
			.547+02	-.104+02	-.636+00	.255+01	-.820+01	.231+03
				.290+01	-.217+01	.127+02	-.126+01	-.434+02
					.731+02	-.419+03	.867+02	-.103+01
						.355+04	-.354+03	.811+01
							.198+04	-.174+01
								.190+04

..... STANDARD DEVIATIONS

	POSITION			VELOCITY			PLATFORM	
1190.9220	294.4005	1429.3466	7.4007	1.7035	8.5502	59.5941	44.5350	43.6678

Table 2.4.2
Shuttle Insertion Covariance and
Correlated Platform Errors

R	T	C	DR	DT	DC
END OF TRACK +3 HOURS					
.15156E+06					
-.13264E+07	.13441E+08				
-.13225E+05	.15304E+06	.24803E+05			
.14617E+04	-.14838E+05	-.17233E+03	.16389E+02		
-.10256E+03	.84288E+03	.87145E+01	-.92948E+00	.73457E-01	
.36138E+01	-.77049E+02	.86978E-01	.83558E-01	-.77959E-03	.36467E-01

SQUARE ROOT OF DIAGONAL

.38931E+03	.36662E+04	.15749E+03	.40483E+01	.27103E+00	.19096E+00
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RSS OF POSITION AND VELOCITY

.3690173068E+04	.4061901100E+01
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$\hat{\underline{R}} = \text{Unit } (\underline{R}_T)$

$\hat{\underline{T}} = \text{Unit } (\hat{\underline{C}} \times \hat{\underline{R}})$

$\hat{\underline{C}} = \text{Unit } (\underline{R}_T \times \underline{V}_T)$

Table 2.4.1
3b Target Covariance

3.0 PARAMETRIC STUDIES

Before undertaking statistical and real-time analysis, it was necessary to determine nominal values of the insertion conditions, braking gates, and delta-v usage. These were found for an envelope of transfer intervals and initial ranges, subject to the general conditions of Section 2.3.

3.1 Pre-rendezvous Events

Technically, the rendezvous began at insertion, where an attempt was made to be on an intercept. It is assumed that no rendezvous activities occur prior to Ins + 5 min., due to post-insertion checks and the tank deorbit operations.

3.2 Duty Cycle/Braking Schedule

A nominal crew duty cycle of 20 percent was chosen as an acceptable compromise between the desire to keep crew workload small and avoiding an overly long extension of the rendezvous time. To ensure a close adherence to the nominal duty cycle, the braking schedule was designed (figure 3.2.1) such that at each discrete range gate the velocity would be reduced by 5 fps. The 5 fps velocity reductions at each gate were a compromise between the number of braking gates and the frequency with which they occurred, assuming a T/M of 1.0 f/s^2 . The range gates were scaled up or down in the digital parametric runs, depending upon the value of T/M used. To avoid applying unnecessary braking maneuvers early in the rendezvous, a closing rate "bleed off" curve (figure 3.2.2) was used to account for the natural tendency of closing rate to decrease on its own until within a few thousand feet.

The parametric runs were made assuming T/M's of 0.10 to 0.40 f/s^2 and wt's-to-go at insertion of 60° , 70° , and 80° of a 140° perfect intercept from a 15.0 n.m. delta-H. The data generated included actual duty cycle, braking delta-v, and time of transfer (table 3.2.1). Note that the actual duty cycle is near 20 percent as long as T/M is 0.25 f/s^2 or greater, but increases as T/M falls below this value.

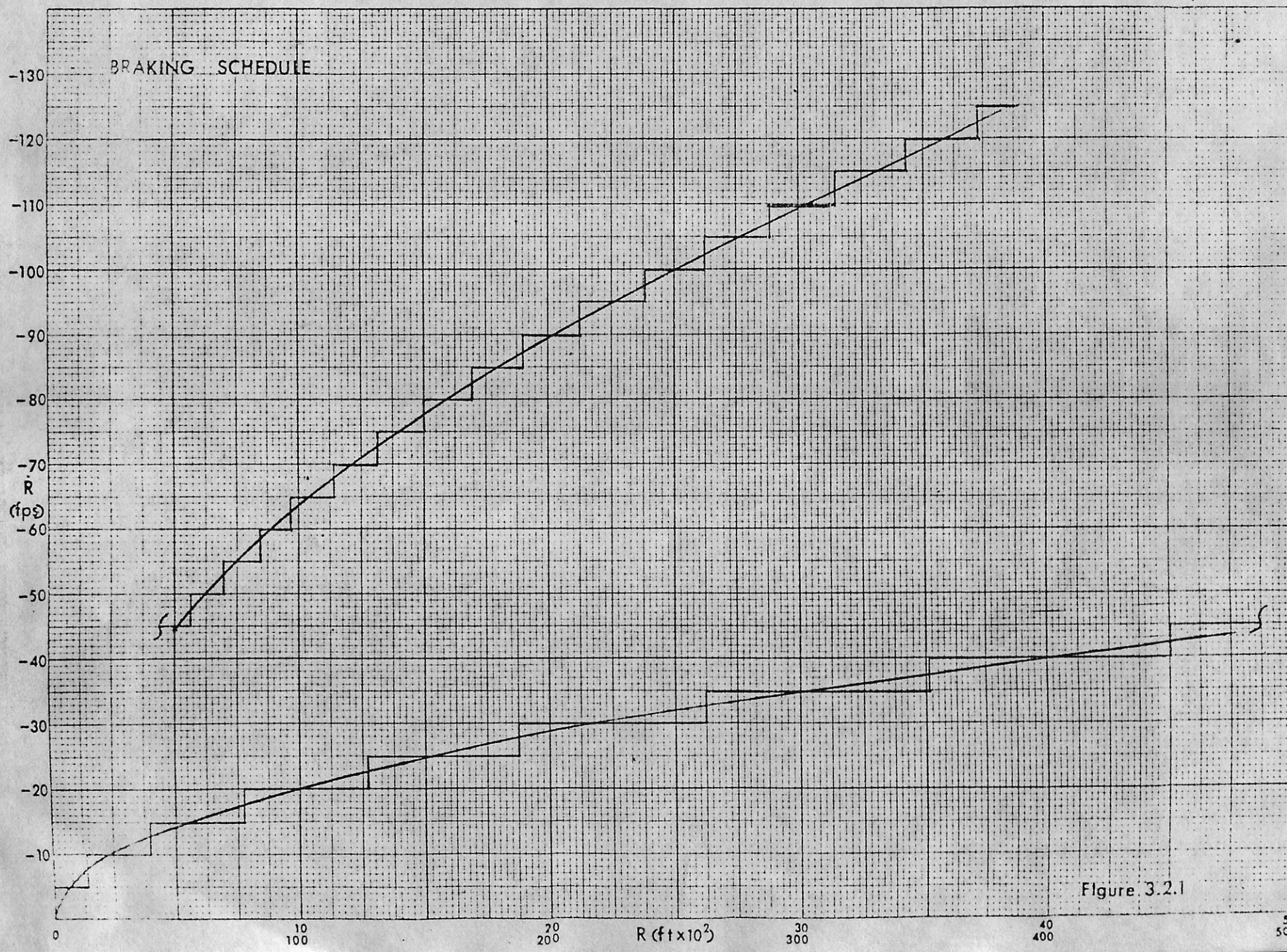


Figure 3.2.1

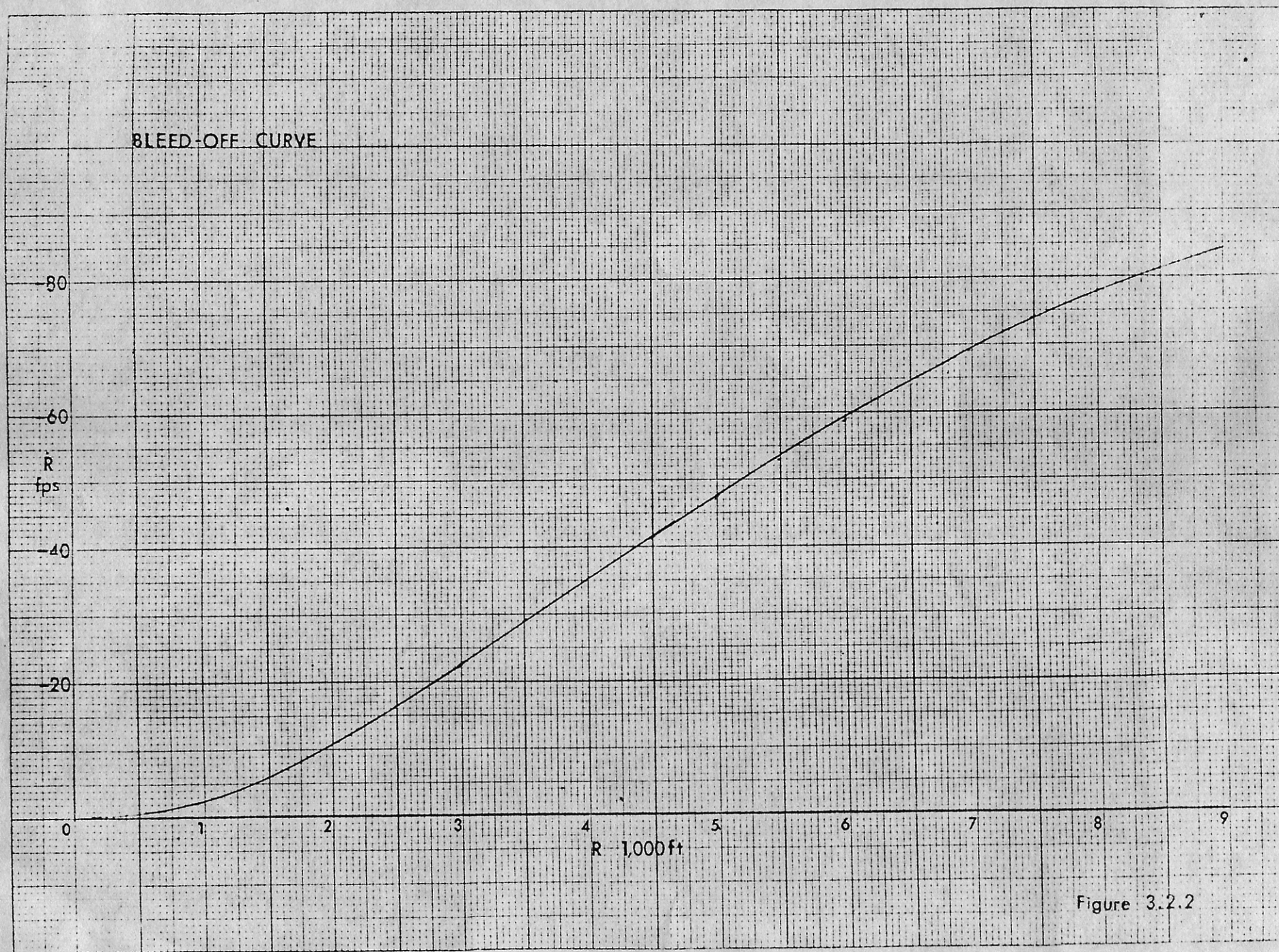


Figure 3.2.2

WT \ T/M	0.10	0.15	0.20	0.25	0.30	0.35	0.40
80°	45.7	44.7	44.8	42.6	43.0	42.7	43.1
70°	46.2	44.6	43.5	42.4	43.8	42.5	42.9
60°	46.1	45.0	45.1	42.8	43.3	43.5	43.0

Table 3.2.1a Delta V (fps)

WT \ T/M	0.10	0.15	0.20	0.25	0.30	0.35	0.40
80°	29.1	25.2	24.5	22.7	22.5	22.4	22.3
70°	33.0	25.5	23.8	22.6	22.9	22.7	22.2
60°	38.7	29.6	25.5	22.9	22.7	22.8	22.8

Table 3.2.1b Duty Cycle (%)

WT \ T/M	0.10	0.15	0.20	0.25	0.30	0.35	0.40
80°	33.5	29.2	26.9	25.5	24.6	23.9	23.4
70°	30.7	26.7	24.4	23.0	22.1	21.2	20.9
60°	27.2	24.2	22.1	20.7	19.8	19.1	18.4

Table 3.2.1c Transfer Time (min)

Table 3.2.1 Various Parameters as a Function of T/M and Wt-to-go

There is the same tendency for the delta-V expended to remain nearly constant ($\approx 42-44$ f/s) until T/M falls below 0.25 f/s^2 (figure 3.2.3). The delta-V expended is not a function of wt-to-go because each wt-to-go represents an insertion onto the same trajectory at a different point in time.

The time of transfer as a function of T/M (figure 3.2.4) demonstrates that for a fixed transfer time there is an wt that minimized the required T/M. For reasons discussed in Section 4.3, the wt selected for the man-in-the-loop runs was 72^0 . For this figure, the minimum T/M would appear to be about 0.24 f/s^2 in order to complete the transfer in 25 minutes. To make allowances for dispersions in intercept time, the design transfer must be less than 25 minutes with a corresponding increase in the T/M requirement to approximately 0.3 (see Section 4.3). The data generated all indicate that T/M's of less than 0.25 f/s^2 incur penalties in delta-V expended and transfer time.

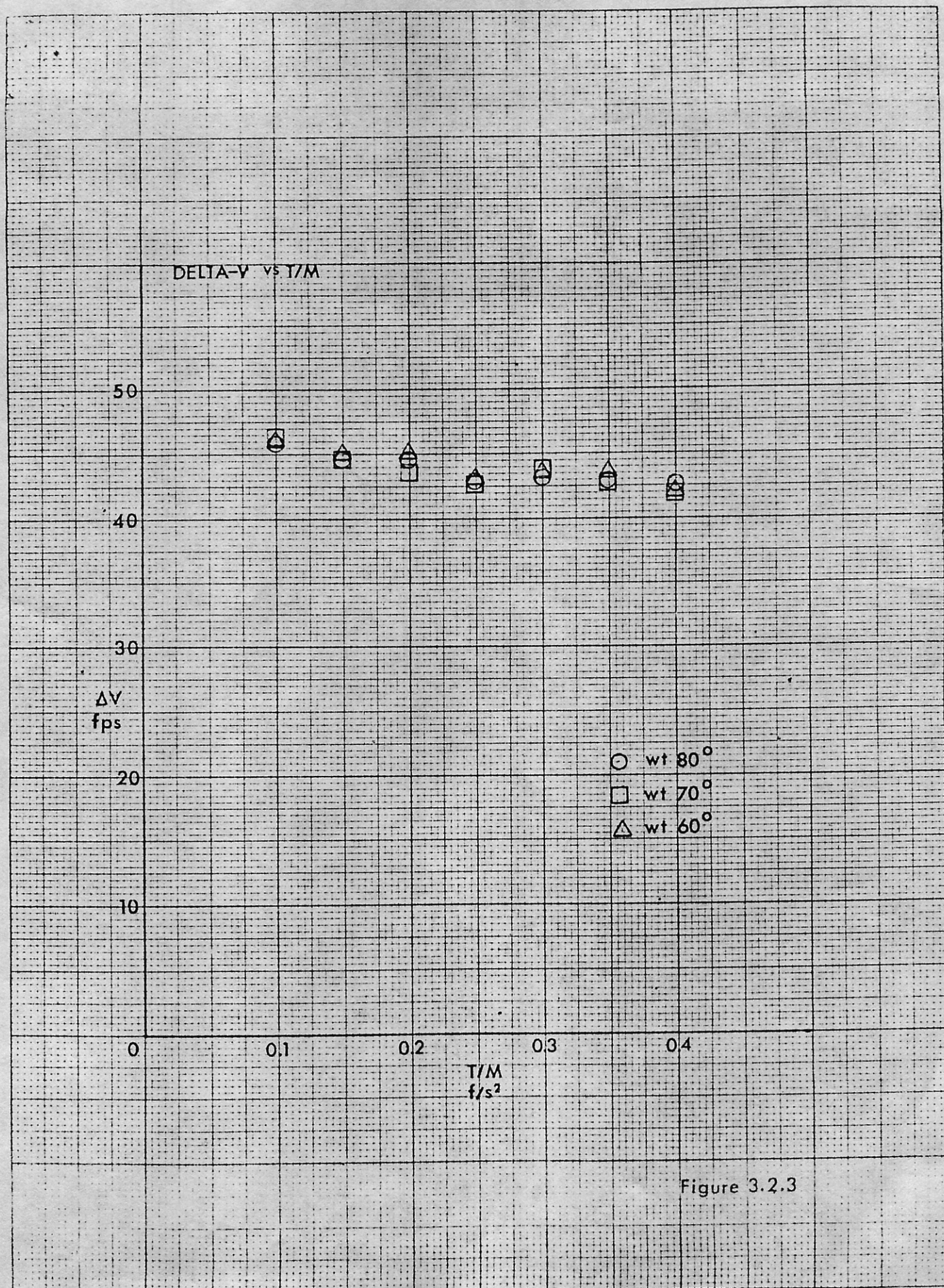


Figure 3.2.3

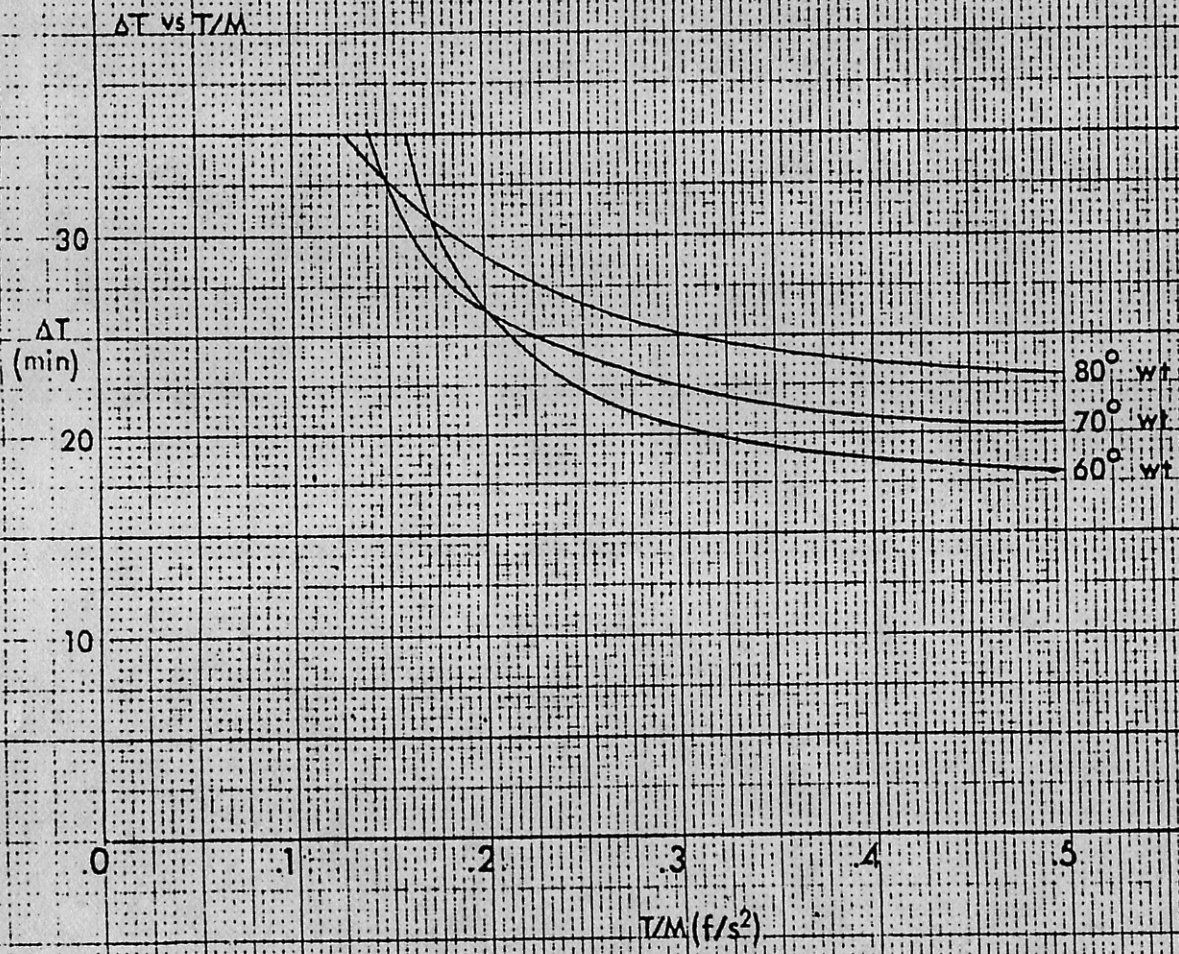


Figure 3.2.4

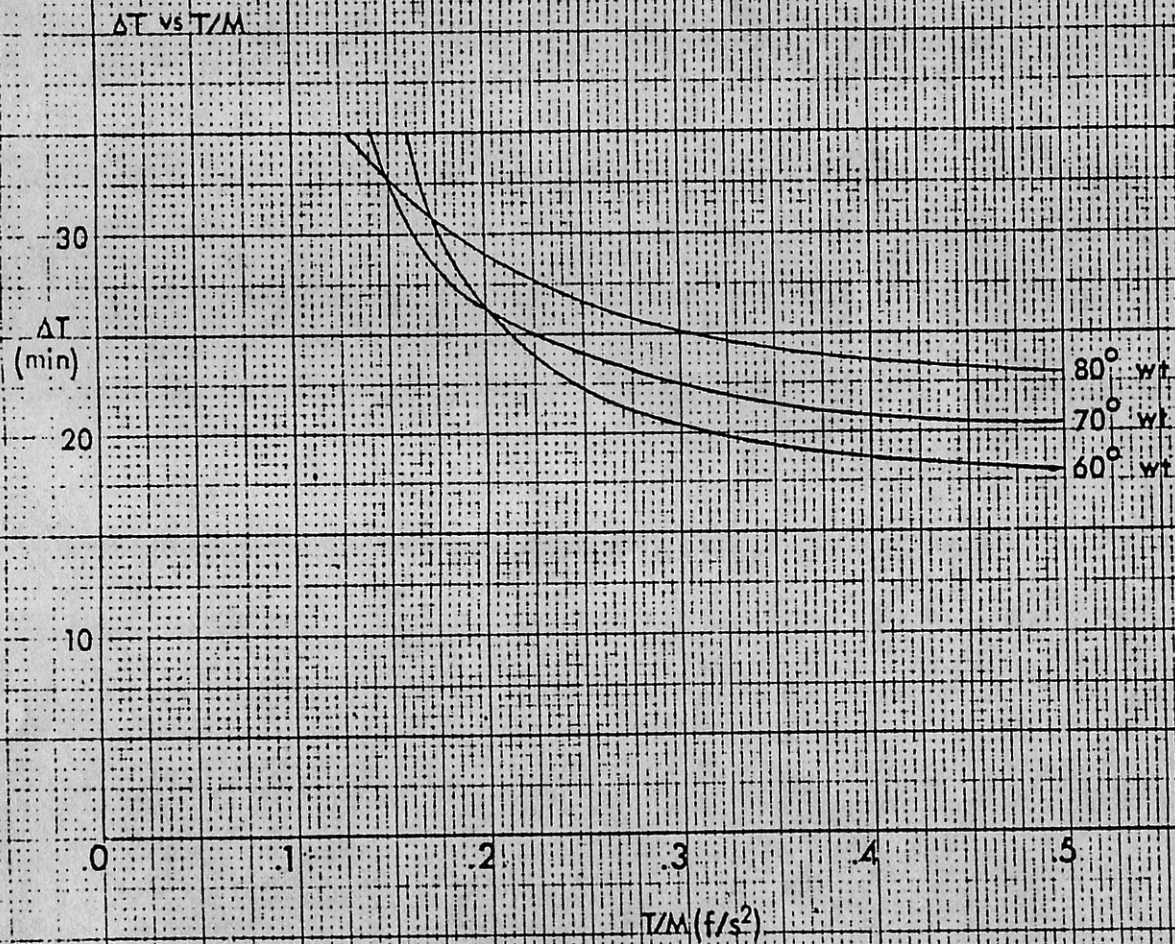


Figure 3.2.4

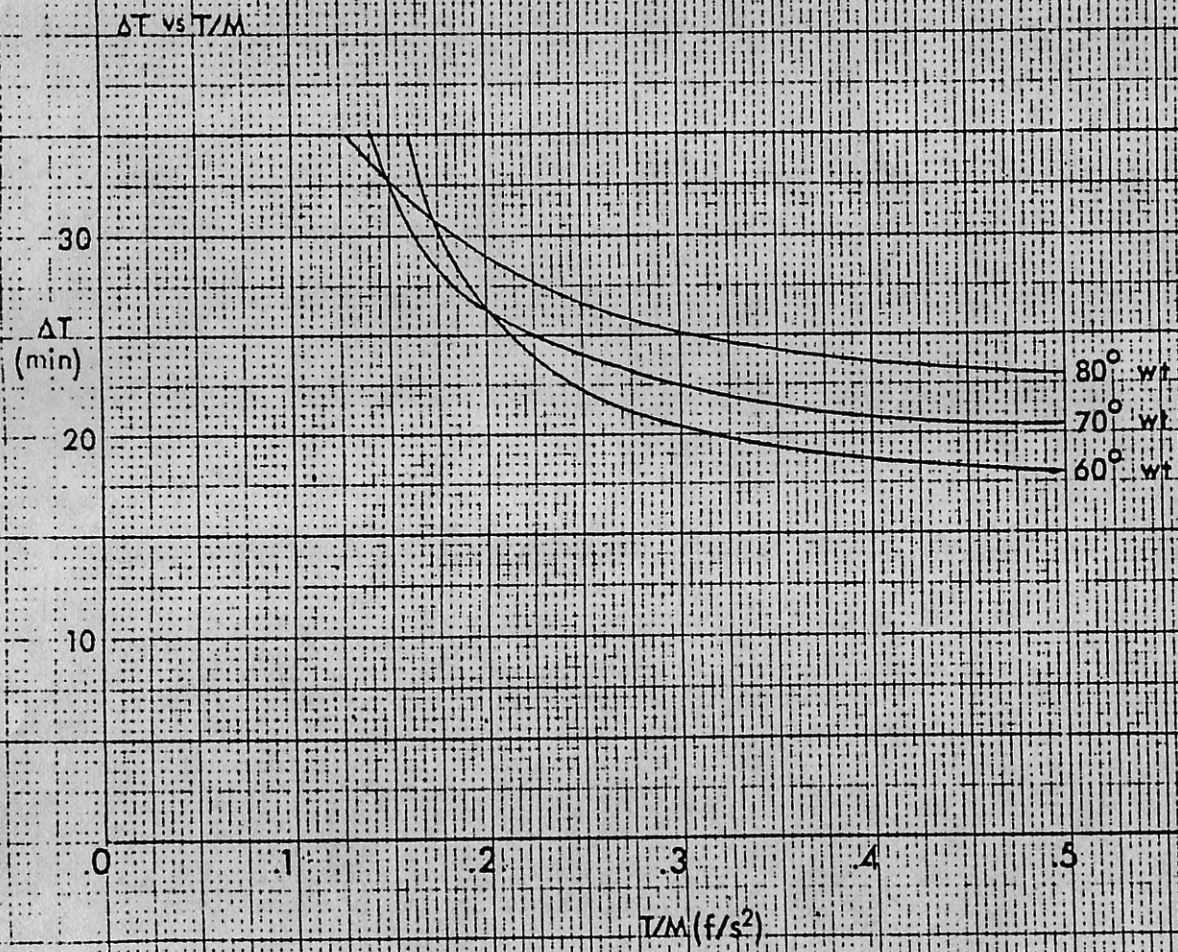


Figure 3.2.4

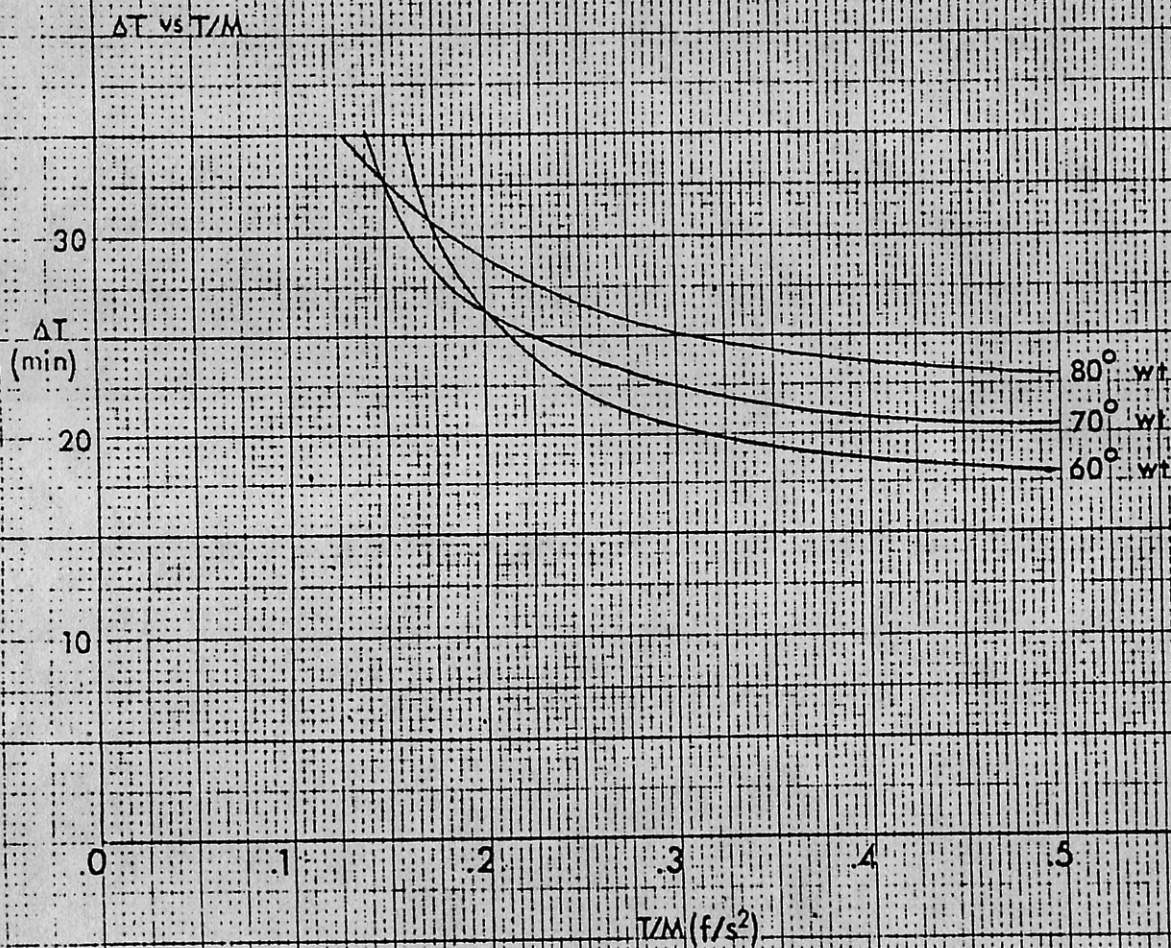


Figure 3.2.4

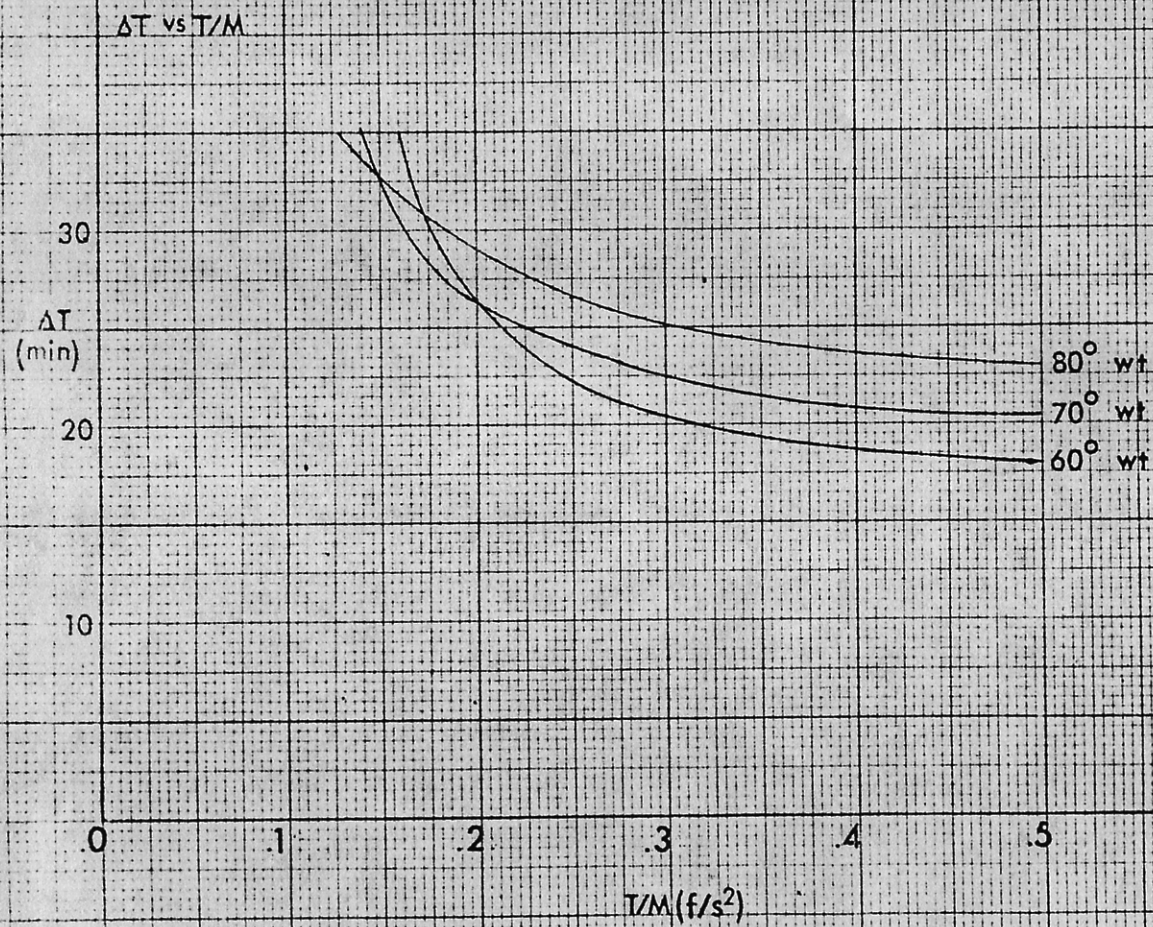


Figure 3.2.4

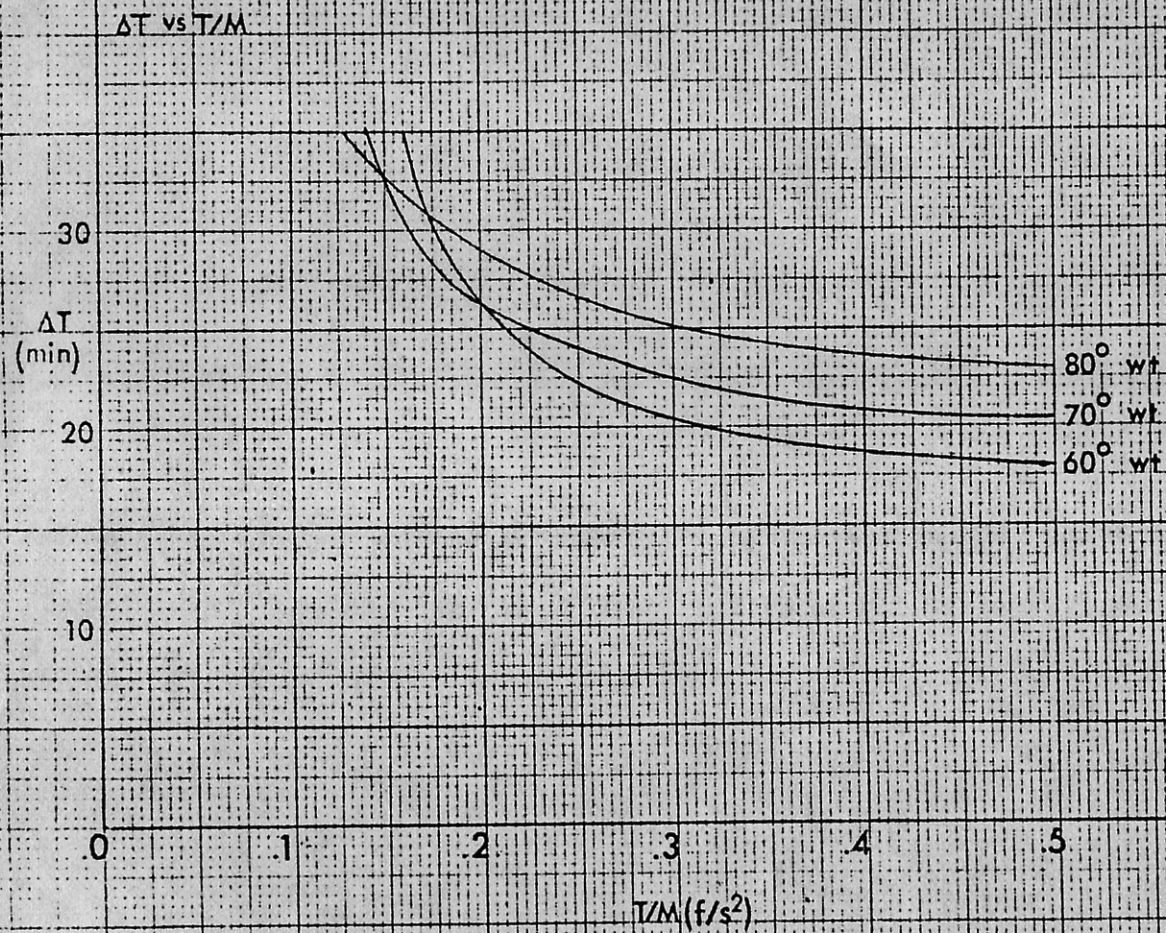


Figure 3.2.4

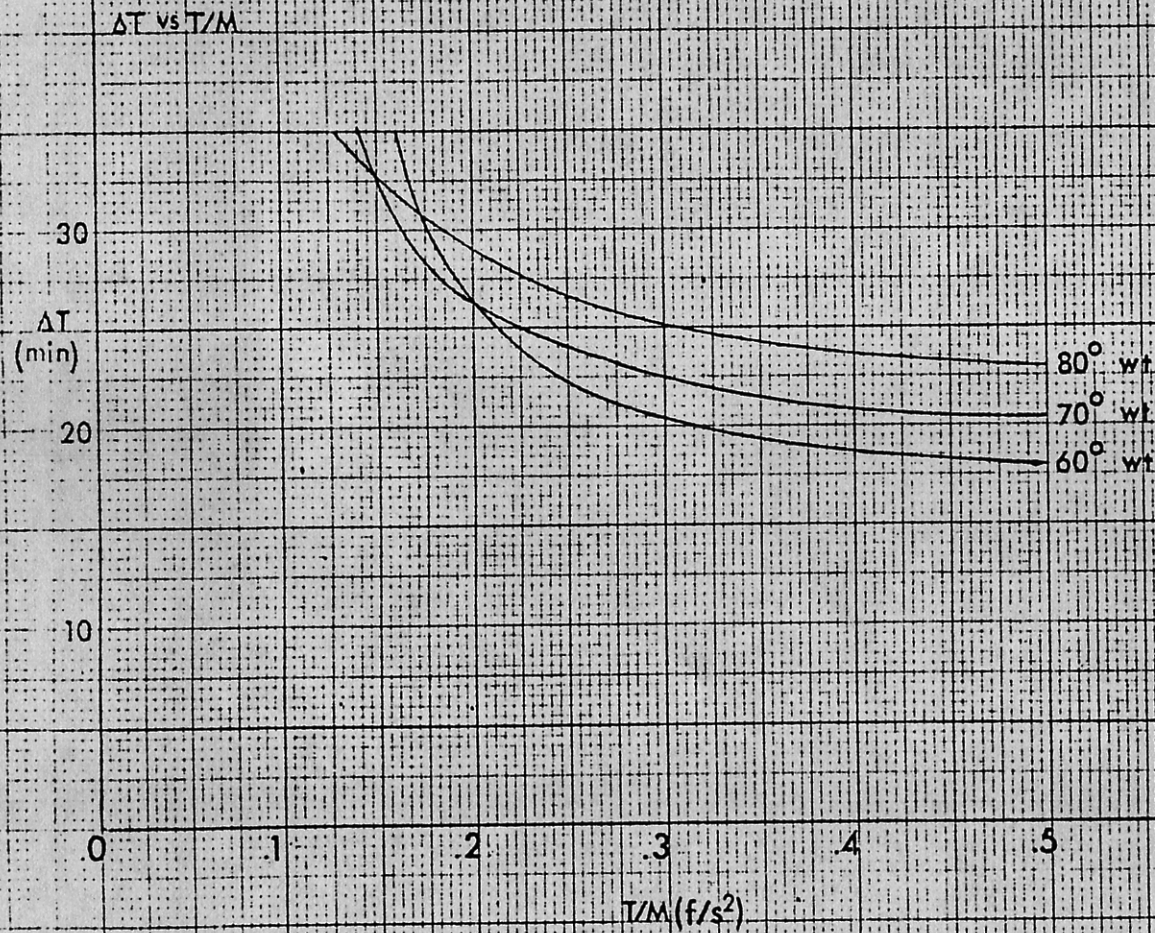


Figure 3.2.4

4.0 MONTE CARLO ANALYSIS

4.1 Program Description

The statistical investigation for Mission 3b was carried out with a three-degree-of-freedom guidance and navigation program. In its cyclic mode, this program generates a set of state dispersions ($X_{ACT} - X_{NOM}$) and errors ($X_{EST} - X_{ACT}$) at the beginning of each cycle. Also modeled are platform misalignments and drift rates, measurement biases and accelerometer scale factor errors. A special braking routine was incorporated to simulate manual performance of the LOSC braking scheme. Estimated and actual states are stored at each maneuver, as well as delta-V's and other parameters as required. At completion of a set, the data were stored permanently on mag tape, from which statistics and data for initializing the SPS were computed. A set consisted of 50 cycles, except where the distribution function of a parameter was being estimated. Experience has shown that 200-300 runs are required to reliably find the "wings" of the distribution.

4.2 Matrix of Cases

The initial goals of the statistical investigation were as follows:

- a. Determine the variance in arrival time, insofar as it is a function of braking scheme, initial dispersions, presence or absence of a midcourse correction, transfer interval, insertion range, and thrust-to-weight (T/W).
- b. Determine the delta-V and duty cycle variations as in (a).
- c. Determine if sufficient time was available for a navigated midcourse before the start of braking.

4.3 Results

For the Monte Carlo analysis, a wt of 72° was chosen as the best common ground between the arrival time, minimum E.T. deorbit range, and braking delta-v constraints. As can be seen in figure 4.3.2, at 72° the nominal E.T. deorbit range is 7.5 nm. In order to maintain this range, it was necessary to let the intercept velocity go up to 42 fps, corresponding to a wt = 130° transfer from 15 nm. delta-h. The effect of varying thrust to mass was assessed with the cases 1 through 4 of table 4.3.1, results of which for total fuel and arrival time are presented in figures 4.3.1a and 4.3.1b. A T/M of 0.3 was chosen for more detailed analysis because higher values do not significantly improve performance in the two critical areas of fuel and arrival time. It is felt that this rather low value of T/M will provide acceptable control authority in all but severe contingency situations. Table 4.3.2 presents the data from 50 cycles at wt = 72° , T/M = 0.3 assuming the covariances of section 2.0. In order to meet the arrival time constraint, it was found necessary to perform a MCC based on raw radar data immediately after acquisition. The equations implemented for this purpose are developed in appendix 7.1.

A further 200 cycles were run to determine the frequency distribution of total delta-v and arrival time. These are presented in figures 4.3.3a and 4.3.3b, and indicate that the 99.6% confidence level for rendezvous delta-v is 150 fps.

For the final set of statistical runs, a dispersion maneuver simulating the effect of off-nominal tank venting and separation delta-v was applied at Ins + 2 min. This was a Gaussian perturbation of mean -2.3 ft/sec and sigma of 1.6 ft/sec. All total delta-v's of table 4.3.2 are the absolute value of each component summed for the entire cycle and then averaged over 50 cycles. This reflects axis-by-axis maneuver application.

MONTE CARLO SUMMARY (T/W = 0.3)

50 Cycles @ wt = 72⁰

R_{INS} = 68 kft, Elev. 55.09⁰

<u>QUANTITY</u>	<u>MEAN</u>	<u>SIGMA</u>	<u>MEAN + 3 - SIGMA</u>
Range at E. T. Deorbit	7.4 n.m.	.64 n.m.	9.3-5.5 n.m.
Range at Acquisition	6.6 n.m.	.64 n.m.	4.7 n.m.
Acquisition Search Radius	0.0 deg.	6.36 n.m.	19.1 deg.
MCC DV (FWD/AFT)	10.3 fps	8.8 fps	36.7 fps
MCC DV (LEFT/RT)	8.5 fps	7.1 fps	29.8 fps
MCC DV (UP/DWN)	11.6 fps	9.5 fps	40.1 fps
TOTAL MCC DV	30.4 fps	15.7 fps	77.5 fps
BRAKING DV (FWD/AFT)	42.8 fps	4.8 fps	57.0 fps
BRAKING DV (LEFT/RT)	2.4 fps	2.8 fps	10.9 fps
BRAKING DV (UP/DWN)	2.8 fps	1.9 fps	8.5 fps
TOTAL BRAKING DV	43.7 fps	4.9 fps	58.5 fps
DUTY CYCLE (FWD/AFT)	30.2 %	2.7 %	38.4 %
DUTY CYCLE (LEFT/RT)	2.0 %	2.0 %	7.7 %
DUTY CYCLE (UP/DWN)	2.0 %	1.3 %	5.9 %
TOTAL DUTY CYCLE	31.0 %	2.9 %	39.6 %
RENDEZVOUS DV (FWD/AFT)	52.8 fps	12.5 fps	90.2 fps
RENDEZVOUS DV (LEFT/RT)	10.8 fps	7.6 fps	33.5 fps
RENDEZVOUS DV (UP/DWN)	14.4 fps	9.1 fps	41.8 fps
TOTAL RENDEZVOUS DV	78.1 fps	17.8 fps	131.4 fps
TIME OF INTERCEPT	21.3 min.	.39 min	22.4 min

Table 4.3.2

MONTE CARLO STUDY

50 RUNS EACH CASE

PLATFORM MISALIGNMENT AND DRAFT

MANEUVER APPLICATION ERRORS: MISALIGNMENT, SCALE FACTOR, GRANULARITY

RADAR MODEL: $\sigma_R = 33$ FT, $\sigma_{\dot{R}} = .43$ FPS, $\sigma_{AZ} = \sigma_{EL} = 2$ MR, $\sigma_{BIAS} = 20$ MR, ANGLE RATE NOISE = .01MR (1σ)

CASE 1: Ins at 68000 ft., MCC at 5 min, wt = 72° , MCC at 5 min, T/M = .1

" 2:	"	"	"	"	"	= .2
" 3:	"	"	"	"	"	= .3
" 4:	"	"	"	"	"	= .4

CASE 5: SAME AS CASE 3, BUT 200 CYCLES FOR FREQUENCY DISTRIBUTION

MONTE CARLO STUDY MATRIX

Table 4.3.1

TOTAL RDZ DELTA-V
fps

$\mu + 3\sigma$

μ

$T/M \text{ fps}^2$

FIGURE 4.3.1a

TIME OF INTERCEPT
min

$\mu + 3\sigma$

μ

$T/M \text{ fps}^2$

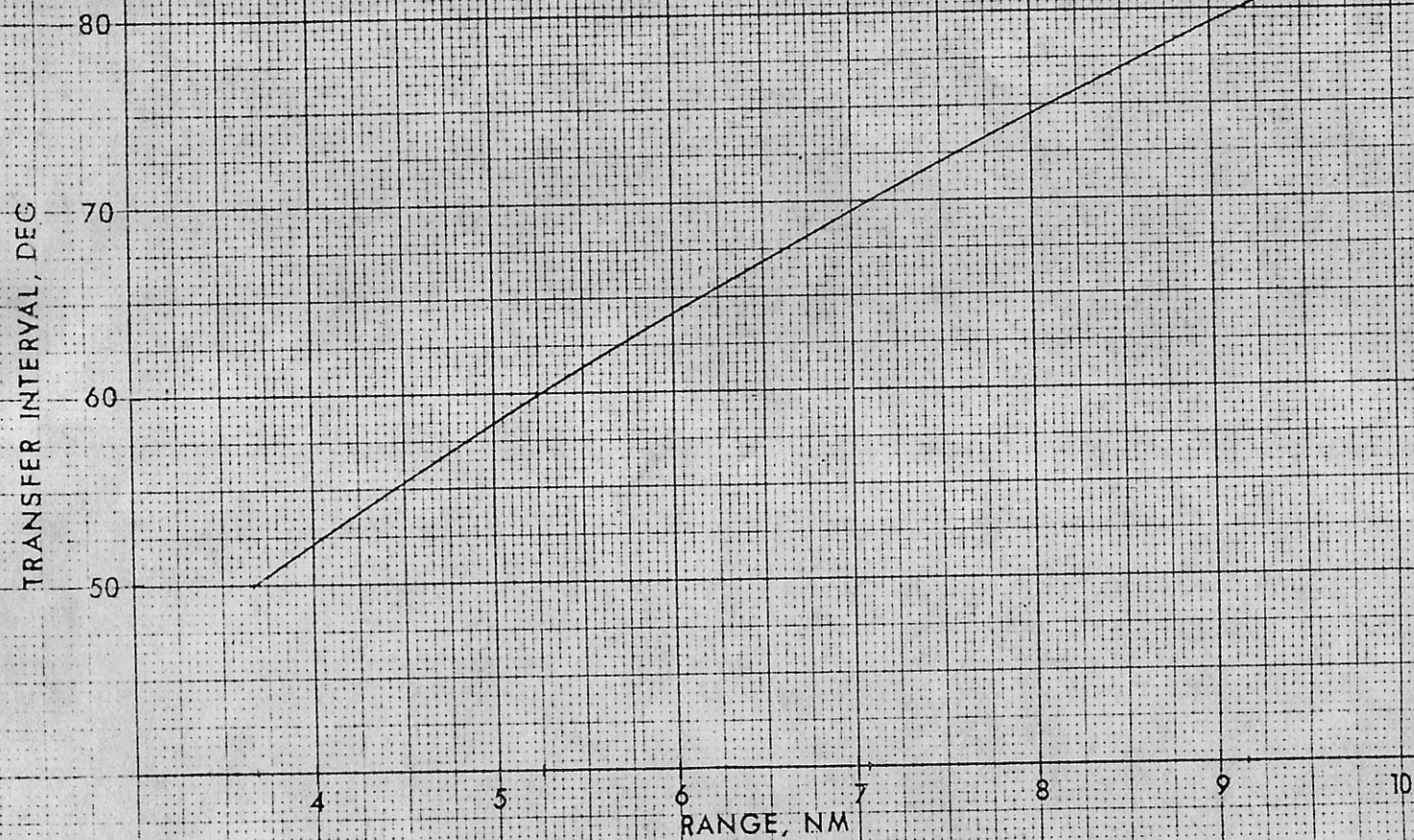
FIGURE 4.3.1b

FIGURE 4.3.2

RANGE AT H/O TANK DEORBIT
AS A FUNCTION OF WT

CLOSING RATE = .42 FPS

APPROACH ANGLE = 135 DEG



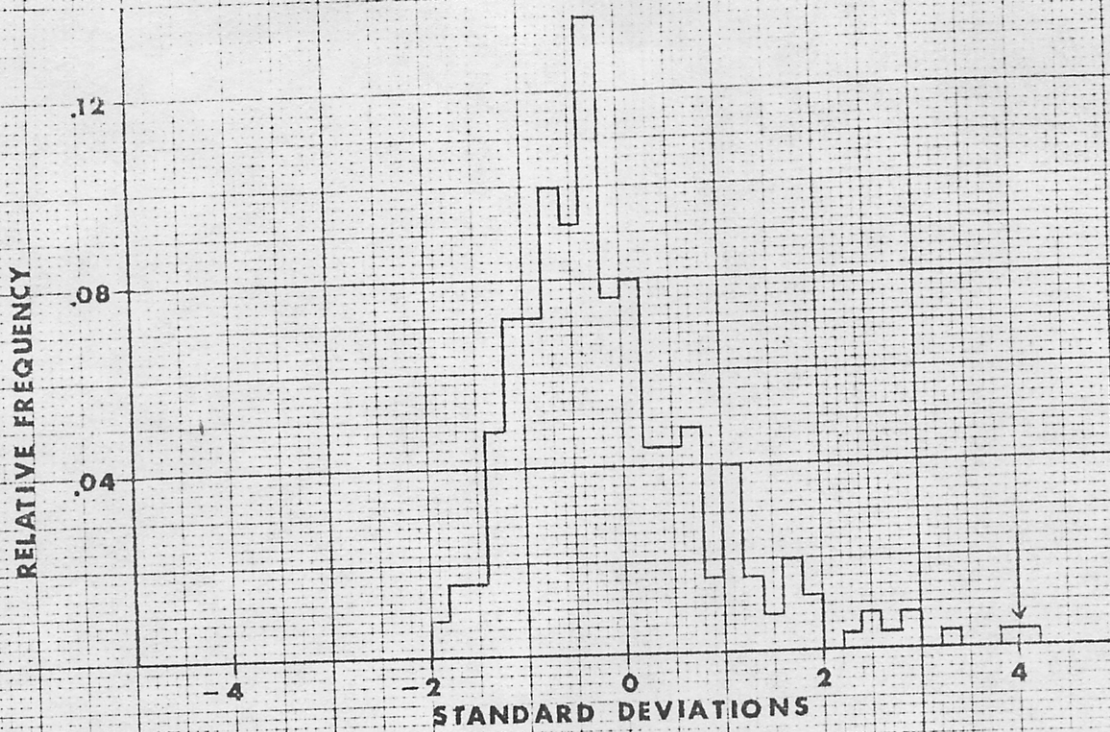


FIGURE 4.3.3a TOTAL RDVZ DELTA-V

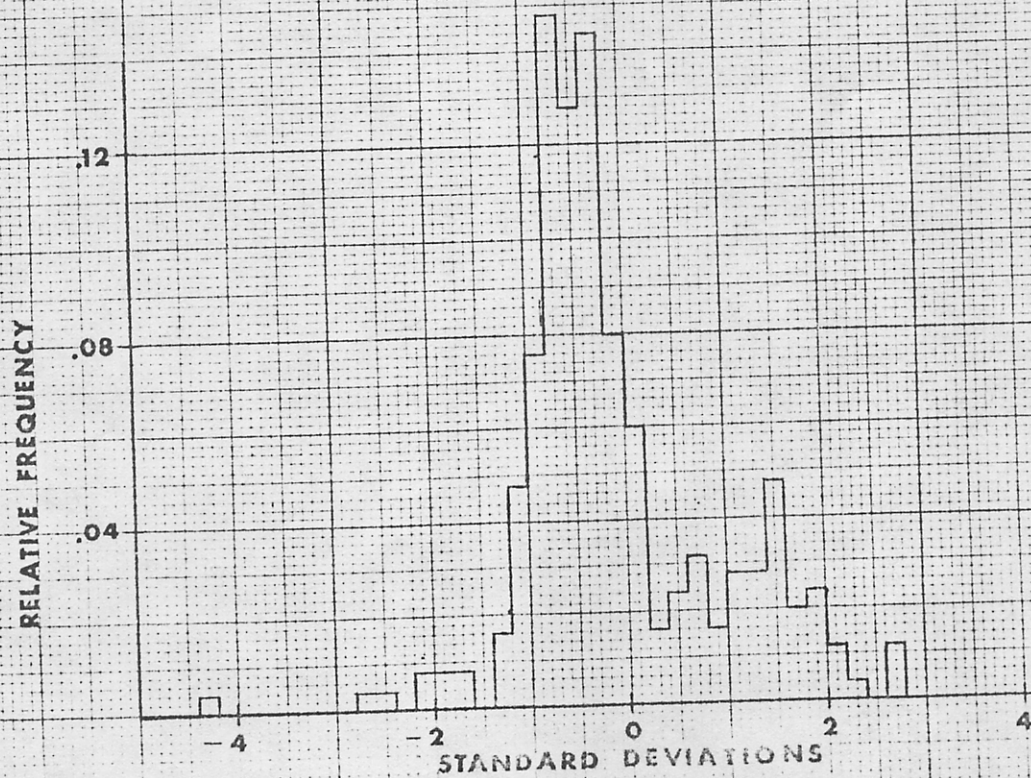


FIGURE 4.3.3b TIME OF INTERCEPT

5.1 Simulator Description

The real-time man-in-the-loop simulations were conducted on the Shuttle Procedures Simulator (SPS) by subjects from Crew Procedures Division, the Astronaut Office, and the Air Force. The SPS simulated the NAR0007D Shuttle Vehicle and consisted of an SSV crew station with a visual system capable of displaying earth, star field, and rendezvous target scenes mounted in the pilot's window. The crew displays and instruments included an FDAI, Range/Range Rate Tapemeter, LOS Rate Crosspointers, and a keyboard with 3 digital readouts to provide crew interface with the guidance computer. The sensors simulated were an optical tracker with a 30° half-cone field-of-view mounted along the SSV + X body axis, which provided angular data, and an omnidirectional range/range rate sensor to drive the tapemeter displays. The mass properties of the assumed configuration are contained in table 5.1.1. The control system modes included a rate command/attitude hold feature with selectable rates and deadbands as well as a hybrid control mode having minimum impulse in the pitch and yaw axes and attitude hold in the roll axis. This mode proved valuable to the subjects in minimizing the pilot workload and attitude propellant, as target tracking could be accomplished efficiently in pulse using pitch and yaw inputs, while little attention was needed to maintain roll attitude.

5.2 Initial Conditions

The nominal initial conditions of the SPS are those of table 4.3.2. The data from the final Monte Carlo set were processed at $\text{Ins} + 5 \text{ min.}$ to obtain a mean vector and a covariance of dispersions. The selection of case a was performed by entering the appropriate initializer for the pseudo-random number generator at the beginning of the run. The initial vehicle attitude was roll = 180° , pitch = 45° , yaw = 0.

The 10 test cases presented in table 5.2.1 were selected from a final Monte Carlo set of fifty cycles. They reflect the earliest and latest arrival times, lowest and highest delta-v, and various other characteristics typical of the entire envelope.

			BASELINE	50% BASELINE
MASS		(lb)	213890.	213890.
C. G. LOCATION: X		(in)	1082.	1082.
	Y	(in)	0.	0.
	Z	(in)	381.	381.
MOMENT OF INERTIA:	I_{XX}	(sl ft ²)	$821. \times 10^3$	$821. \times 10^3$
	I_{YY}	(sl ft ²)	$7130. \times 10^3$	$7130. \times 10^3$
	I_{ZZ}	(sl ft ²)	$7376. \times 10^3$	$7376. \times 10^3$
PRODUCE OF INERTIA:	I_{XY}	(sl ft ²)	$-7. \times 10^3$	$-7. \times 10^3$
	I_{XZ}	(sl ft ²)	$207. \times 10^3$	$207. \times 10^3$
	I_{YZ}	(sl ft ²)	$0. \times 10^3$	$0. \times 10^3$
ACCELERATION:	X	(f/s ²)	0.60	0.29
	Y	(f/s ²)	0.45	0.22
	Z	(f/s ²)	0.90	0.45

SPS Orbiter Mass Properties

Table 5.1.1

SIMULATOR CASE #	DELTA-V	TIME OF INTERCEPT	DUTY CYCLE	REMARKS	RUN #
1	66.2	21.1	32.0	Lowest Delta-V	17
2	74.1	20.9	33.2	Small MCC	13
3	81.3	21.3	28.7		18
4	84.8	21.2	36.7	High Duty Cycle Largest MCC Out of Plane	31
5	99.9	20.4	28.2	Earliest Arrival Time	22
6	90.3	23.3	28.1	Late Arrival Time Speed-Up Required at MCC	12
7	110.8	21.8	35.9	High Duty Cycle	39
8	124.2	24.1	25.0	Latest Arrival Time	29
9	150.0	21.2	38.2	Highest Braking Out of Range	9
10	156.7	22.0	42.0	Largest Delta-V, Duty Cycle	24

REAL-TIME SIMULATOR CASES

TABLE 5.2.1

5.3 Matrix of Runs

The man-in-the-loop simulations were conducted using 3 subjects who completed a matrix of 30 runs each. The 30 runs consisted of 3 runs each of the 10 dispersed initial conditions discussed in Section 5.2. The first set of runs were completed with the aid of the Range/Range Rate Tapemeter, pointing information, and LOS Rate meter. To simulate the effects of a sensor configuration that provides only ranging, pointing, and range rate data, the runs were repeated without the aid of the LOS Rate meter. This required the pilot to estimate the LOS rate corrections by boresighting on the target and noting the amount of angular drift accrued over a one minute time period, causing an increase in attitude fuel consumption and uncertainty in LOS delta-v. To simulate a sensor which provides only ranging information, a third set of runs were made without the aid of either the LOS Rate meter or Range Rate Tapemeter. In addition to estimating the LOS rate corrections, as in the second set of runs, the pilot differenced the range at 50 second intervals to estimate the closing rate. Again, the necessity for tracking the target resulted not only in increased attitude fuel consumption and LOS delta-v uncertainty, but the need to estimate closing rate with its associated uncertainty and delay caused a severe impact on crew workload.

In those cases where the LOS rate meter was available, a midcourse correction was computed at Insertion + 4 minutes to preserve the time of arrival. The midcourse consisted of a chart with inputs of range and target pitch and yaw angles relative to local horizontal. The chart outputs were desired range rate and desired elevation and azimuth rates. For the runs without the LOS rate meter, the pilot estimated the LOS rates and kept them near zero while adjusting the closure rate to ensure a timely intercept.

After the midcourse had been applied (or immediately if no midcourse solution was used), the pilot monitored range rate and LOS rates and applied corrections as required to maintain the braking schedule.

A run was considered terminated when the pilot had achieved near zero rates at a range of approximately 50 feet.

In addition to the matrix of 30 runs described, two of the subjects completed a matrix of 15 additional runs. To simulate an SSV with lower T/M (table 5.1.1) the output of each thruster was reduced from 1000 lbs to 500 lbs and each of the last 5 I.C.'s of the 10 I.C.'s available were rerun, first with all available displays, then without the crosspointers, and finally with ranging data only.

5.4 Results

The results of the first matrix of runs are illustrated in table 5.4.1, with cases 1 through 3 representing the successively degraded display configurations. The parameters of interest included:

- a. Arrival Time - the time after Insertion at which the rendezvous was completed.
- b. First Maneuver - the time after Insertion at which the first braking maneuver commenced.
- c. X-Duty Cycle - the percentage of time in the rendezvous spent thrusting along the vehicle-X axis.
- d. Y-Duty Cycle - the percentage of time in the rendezvous spent thrusting along the vehicle Y axis.
- e. Z-Duty Cycle - the percentage of time in the rendezvous spent thrusting along the vehicle Z axis.
- f. Attitude Propellant - the amount of propellant expended for attitude control.
- g. Translation Propellant - the amount of propellant expended for translation.
- h. Total Propellant - the sum of attitude propellant and translation propellant.
- i. Tank A - the amount of propellant expended for attitude control and translation from the nose tank.
- j. Tank B - the amount of propellant expended for attitude control and translation from the right rear tank.

k. Tank C - the amount of propellant expended for attitude control and translation from the left rear tank.

The table lists, for each case, the mean value and mean + 3 value for each of the parameters.

Since one of the groundrules of the 3B mission is that the rendezvous shall be accomplished in 25 minutes or less, it is interesting to observe that this criterion is met by those runs where the LOS Rate meter was available (indicating that the midcourse correction did indeed preserve the arrival time) but was not met by the succeeding cases. In fact, the ranging only case exceeded the mean + 3 allowable time by some 3.47 minutes.

The time at which the first braking (range rate reducing) maneuver occurred was important because the duty cycle was computed over the time span between the first braking maneuver and the time of arrival. If corrections normal to the line of sight were necessary before a braking maneuver was required, the time for computing the Y and Z axis duty cycles was initiated with the first normal correction (otherwise, it was initiated at the time of the first braking maneuver).

The mean value for the duty cycle in the X axis in all cases is approximately 15 percent. The largest effect of the degraded display modes is to force an increase in the mean + 3 X duty cycle. A more dramatic increase can be observed in the Y and Z duty cycles (both mean and mean + 3) for the degraded display modes. This increase is primarily due to the increased uncertainty in estimating the LOS rates and nulling them. A less tangible effect of the reduction in sensor capability, and one not reflected in the duty cycle calculations, is the increased load on the pilot due to the requirement to monitor LOS angle to estimate LOS rates and range to estimate range rate. Pilot comments relative to the workload for each display and T/M situation are contained in Section 5.4. The data for propellant consumption for attitude control and translation indicate the same trend for a decreased display capability to be accompanied by an increased propellant penalty.

		CASE 1		CASE 2		CASE 3	
		MEAN	MEAN + 3 σ	MEAN	MEAN + 3 σ	MEAN	MEAN + 3 σ
ARRIVAL TIME	(min)	21.24	24.87	21.69	26.47	22.18	28.47
FIRST MANEUVER	(min)	12.94	18.56	12.72	19.80	13.52	23.61
X DUTY CYCLE	(%)	14.6	18.3	13.8	19.2	14.7	23.5
Y DUTY CYCLE	(%)	2.4	6.7	7.0	15.3	7.4	18.0
Z DUTY CYCLE	(%)	1.8	5.0	4.5	13.2	4.7	14.0
ATTITUDE PROPELLANT*	(lb)	377.	737.	540.	1021.	531.	1036.
TRANSLATION PROPELLANT	(lb)	3035.	5623.	3544.	6902.	3614.	7062.
TOTAL PROPELLANT	(lb)	3411.	6132.	4084.	7554.	4144.	7821.
TANK A	(lb)	2077.	3174.	2348.	3688.	2357.	3767.
TANK B	(lb)	591.	1382.	804.	1848.	833.	1977.
TANK C	(lb)	743.	1958.	932.	2335.	953.	2415.

*The propellant numbers reflect only the post ET deorbit to stationkeeping mission phases.

Summary of 3b Studies on SPS Using Baseline Orbiter

Table 5.4.1

		CASE 1		CASE 2		CASE 3	
		100% T/M*	50% T/M	100% T/M	50% T/M	100% T/M	50% T/M
ARRIVAL TIME	(min)	21.72	22.19	22.24	22.80	22.92	23.03
FIRST MANEUVER	(min)	13.44	14.51	13.39	15.80	15.42	15.59
X DUTY CYCLE	(%)	15.4	33.2	14.7	32.0	16.2	29.6
Y DUTY CYCLE	(%)	3.0	4.7	8.2	17.9	8.4	17.1
Z DUTY CYCLE	(%)	2.1	4.8	6.0	13.3	6.5	13.2
ATTITUDE PROPELLANT	(lb)	412.	380.	573.	614.	585.	719.
TRANSLATION PROPELLANT	(lb)	3679.	3509.	4356.	4225.	4390.	4544.
TOTAL PROPELLANT	(lb)	4091.	3889.	4929.	4839.	4975.	5262.
TANK A	(lb)	2268.	2138.	2603.	2436.	2581.	2630.
TANK B	(lb)	795.	785.	1068.	1136.	1100.	1228.
TANK C	(lb)	1026.	965.	1256.	1267.	1293.	1404.

* The higher mean propellant usage relative to table 5.4.1 is due to the five most difficult I.C.'s being selected.

Summary of 3b Studies on SPS with Baseline and 50% Baseline Orbiter

Table 5.4.2

The rise in the attitude propellant consumption is directly attributable to the requirement to track the target to estimate LOS rates. The rise in translation propellant can be attributed to the increased uncertainty in the LOS rates and, for the ranging only case, some uncertainty in the closing rates. The data for total propellant consumption are included as well as the distribution by tank. Although the indicated mean + 3 propellant usages fall within the amounts available for each tank, it should be noted that most is used from the nose tank (tank A) and the pad (excess) is very slim for those cases without a full complement of display data. A more equitable distribution among the tanks may be achieved by using some other vehicle axis (- Z for instance) for braking.

Table 5.4.2 compares the results of the second matrix of runs using an orbiter with 50 percent of the T/M of the baseline vehicle. Since each case consisted of only 5 I.C.'s run by 2 subjects, only mean values of the parameters of interest are listed. Also, to be consistent, the mean values for the data obtained for the 100 percent T/M orbiter were generated from the same 5 I.C.'s. These I.C.'s were selected because of their relative difficulty and the insight they might provide into problems associated with a reduction in T/M.

The propellant numbers are fairly consistent for both high and low acceleration with the latter being slightly more efficient with a full complement of display data and slightly less efficient when only ranging data were available. The most significant impact of the lower T/M is in crew duty cycle where in almost every case the resultant duty cycle is at least doubled. Assuming that all braking maneuvers, both along and normal to the line of sight, were accomplished sequentially, the crew duty cycle is up to 60 percent on the average for both degraded display modes. Assuming that the data from the earlier study can be extrapolated to include the lower T/M by doubling the values for duty cycle, it can be seen that the mean + 3 duty cycle (assuming sequential corrections) for the case of range/range rate data only approaches 95 percent and actually exceeds 100 percent for the case of ranging data only.

For the case where a full complement of sensor data (range, range rate, and LOS angle rates) is available, the mean + 3 duty cycle is 60 percent.

5.4 Pilot Comments

This section contains the comments of simulator subjects A and B relative to the performance of the mission 3b rendezvous for the three sensor display modes in both the baseline and the 50 percent baseline configurations. An evaluation of each set of runs in tabular form is included.

5.4.1 Subject A

I served as an evaluation pilot for a Shuttle Mission 3b rendezvous simulation on the MSC Shuttle Procedures Simulator from December 1972 through January 1973. This letter provides pilot commentary to compliment the simulation data.

The simulator provided range, range rate, angle and angle rate displays. The star field and target vehicle were displayed visually, and attitude and translation control modeled a recent NR orbiter configuration, controlled through hand controllers. Although the displays, controls, and simulator software were quite adequate to investigate the rendezvous phase, no other vehicle subsystems or crew procedures were included. This was a part-task simulation of rendezvous only, and any other crew duties would be additive to the rendezvous worked.

I flew 45 runs for data. The first 10 runs were flown to station-keeping from 10 different insertion conditions with all the displays operative. The line-of-sight (LOS) rate needles were then failed and the 10 runs were repeated. The third set of 10 runs was flown with both LOS rates and range rate failed. Following these runs, the RCS T/W was cut in half and each of the three display configurations was flown again five times.

With all displays available and full RCS T/W, the rendezvous could be easily flown by one man. Pilot workload was relatively low and I felt that I could have been doing some subsystems monitoring in addition to flying the rendezvous.

		CASE 1	CASE 2	CASE 3
THRUST TO WEIGHT	BASELINE	2 Low work load comfortably in control, very minor deficiencies	5 1/2 Controllable, major deficiencies require considerable compensation for reduced performance.	6 1/2 Same as no LOS-rate.
	50% BASELINE	5 Adequate performance with moderate compensation, no margin for failures.	9 Controllability in question. Intense compensation required.	10 Intense compensation required. Inadequate to guarantee control.

TABLE 5.4.1
SUBJECT A EVALUATION

I felt comfortably in control of the situation all the way in and felt that I could have endured limited display or thruster failures and still completed the mission. It was a routine task.

With the LOS needles failed, the rendezvous became a very busy, one-man task. I had no free time to do other monitoring chores. I wasn't always able to determine that the trajectory was on a good terminal phase, as the increased fuel consumption showed, but I was satisfied that I was safe all the way in. There was no collision potential. Further deletion of the range rate display added slightly to my workload, but resulted in the same general comments. I was far more sensitive to loss of LOS rates than range rate. Executing the rendezvous in a controlled fashion required that I bring the LOS rates to zero (or nearly so) and stabilize them there before reaching the first braking gate. When the T/W was reduced, the effect of less "Y" and "Z" thrust was to increase the time required to null LOS rates and raise the nagging fear that I would start piling braking gates up with LOS rate control burns, a situation which could lead to loss of control. The reduced "X" thrust aggravated the situation, since the gates were more time consuming when they did arrive. As long as the LOS rate needles were available, this did not happen and control was not lost; but I felt that any failure, anomaly or distraction, would have led to a wave-off. This rendezvous configuration was not fault-tolerant. Many things have to go exactly right. The low T/W configuration, combined with loss of LOS rates was a frantically busy, harried, hurried, fuel-wasting hazard to flight. I did not, in general, have adequate control over the trajectory and I would not bet that I could accomplish the mission on a routine basis.

The preferred configuration was clearly the nominal display combined with nominal RCS T/W. This configuration appeared to have some margin to accommodate later adverse developments in G&N accuracy, more time-consuming housekeeping procedures, or likely failure states. Although some of the other configurations might currently appear to do the job, they require us to start out this program with our backs already up against the wall.

5.4.2 Subject B

I participated in approximately 50 mission 3b rendezvous simulations on the SPS during December 1972 and January 1973. These were in both the high and low thrust configurations and with full instrumentation as well as with line-of-sight rates and range rates omitted.

. From a pilot's point of view, the configurations with the full instrumentation were the easiest to fly. Omitting the range rate and line-of-sight rates added significantly to the pilot workload until in the low thrust case the pilot and co-pilot were almost totally dedicated to performing the mechanics of determining what were the line-of-sight rates and range rates, and performing the burns. It is my opinion that the timeline during this portion of the mission will be extremely busy. The crew will be checking the vehicle after boost, as well as getting ready for the orbital operation. It would be foolish to require the crew to perform this type of task at that time and not have them available for other, yet undefined or unexpected, tasks that may be required at this time.

The displays used for the 3b mission were adequate, simple to read, and made the rendezvous an easy task. Of course, this changes when the line-of-sight rates and range rates were eliminated. Without these pieces of information, the pilot task was complicated.

The controls employed in the simulation could be significantly improved for the 3b mission. The selection of rate command or minimum impulse required the use of the keyboard and this often proved to be too time consuming to make the switching back and forth a reasonable method of control. In rate command, it was difficult to hold a particular rate simple because it was difficult to hold your hand steady for the required period of time. The minimum impulse mode was not a reasonable system because the pulse was so small that it took an unreasonably large number of pulses to adequately control the vehicle.

The procedures were quite straight forward, but I believe some further investigation into the effects of large corrections would be in order since these vary the preplanned trajectory quite a bit.

		CASE 1	CASE 2	CASE 3
THRUST TO WEIGHT	BASELINE	<u>Easiest to fly</u> Pilot has time to scan other instruments in the cockpit and probably monitor the rendezvous with backup charts.	Pilot is busy timing line of sight rates, but can handle the job comfortably. Little time is available to do anything else. Braking gates may be omitted to allow timing of rates.	Pilot is dedicated to the mechanics of timing line of sight rates, range rates, and doing burns. Pilot can miscalculate rates and over or under compensate.
	50% BASELINE	<u>Easy to fly</u> Burns become longer and in more widely dispersed cases burns may be a couple of minutes long. Pilot still is not completely dedicated to mechanics of the rendezvous.	Pilot is busy timing line of sight rates. Little time is available to do anything else. Braking gates may be omitted to allow timing of rates.	Pilot is completely dedicated to the mechanics of timing line of sight rates, range rates, and doing burns. Outside help is welcome to help do the bookkeeping. Pilot can miscalculate rates and over or under compensate. Some braking gates are missed to allow timing of rates.

TABLE 5.4.2
SUBJECT B EVALUATION

6.0 Conclusion

Several conclusions can be drawn from the analyses and simulation reported herein:

a. The 3b mission rendezvous can be successfully accomplished using the baseline orbiter vehicle (NAROOO7D) configuration. For the design trajectory of 72° wt, the arrival time constraint of 25 minutes between orbit insertion and stationkeeping was met with reasonable propellant consumption and crew workload.

b. Rendezvous flight display requirements include vehicle to target range, range rate, pointing angles and LOS angular rates. Loss of angular rate information causes a significant increase in propellant consumption because of the decrease in accuracy of maneuvers normal to the line of sight. Loss of both angular rate and range rate information results in unacceptably high crew workload as continuous effort is required to estimate closing rate as well as LOS rate.

c. A single midcourse correction is required to meet the 25-minute arrival time constraint. Trajectory dispersions are too large to preserve arrival time if maneuvers are limited to braking and LOS control only.

d. The minimum acceptable translational accelerations are as follows:

$\pm X:$	$.3 \text{ FPS}^2$
$\pm Y:$	$.2 \text{ FPS}^2$
$\pm Z:$	$.3 \text{ FPS}^2$

These values are required to complete the rendezvous within limits on arrival time and crew workload.

e. Mission trajectory design, RCS propellant requirements, rendezvous sensor acquisition angles, and RCS acceleration requirements are strongly coupled and dependent on orbit insertion accuracy. This study was performed assuming a high quality inertial system accurately aligned prior to liftoff. If the G&N system performs substantially worse than assumed herein, the above requirements must be reexamined and adjusted accordingly.

f. The X-axis method of rendezvous results in the bulk of RCS propellant drawn from the forward tank. Consideration should be given to utilizing the -Z axis as the primary rendezvous axis if a more balanced distribution is desirable. Studies must be conducted prior to completion of the design of the rendezvous sensor installation, as the tracking axis must coincide with the primary rendezvous axis.

7.0 APPENDICES

7.1 Closed-Form Midcourse Correction

The Clohessy-Wiltshire equations give the delta-v for intercept as

$$\dot{\underline{r}}_0 = \underline{F} \underline{r}_0 \quad 7.1.1$$

where

$$\underline{F} = \frac{1}{\Delta} \begin{bmatrix} 3ct - \frac{4s}{w} & -\frac{6}{w}(1-c) + 3st & 0 \\ -\frac{6}{w}(1-c) + 3st & -s/w & 0 \\ 0 & 0 & \frac{w\Delta c}{s} \end{bmatrix}$$

$$\Delta = \frac{8}{w^2} (1-c) - \frac{3st}{w}$$

$$c = \cos(wt)$$

$$s = \sin(wt)$$

and $\underline{r}_0$ is the initial position in the local-vertical frame. We define the local-vertical frame

$$\underline{X}_{LV} = \text{unit}(\underline{R}_S)$$

$$\underline{Y}_{LV} = \text{unit}(\underline{Z}_{LV} \times \underline{X}_{LV})$$

$$\underline{Z}_{LV} = \text{unit}(\underline{R}_S \times \underline{V}_S)$$

$$\underline{G}_{LV} = [\underline{X}_{LV} \quad \underline{Y}_{LV} \quad \underline{Z}_{LV}]$$

Also, a line-of-sight frame

$$\underline{X}_{LOS} = \underline{G}_{LV}^T \text{unit}(\underline{R}_T - \underline{R}_S)$$

$$\underline{Y}_{LOS} = \underline{Z}_{LOS} \times \underline{X}_{LOS}$$

$$\underline{Z}_{LOS} = \text{unit}(\underline{Z}_{LV} \times \underline{X}_{LOS})$$

$$\underline{G}_{LOS} = [\underline{X}_{LOS} \quad \underline{Y}_{LOS} \quad \underline{Z}_{LOS}]$$

Then $\underline{r}$ equation 7.1.1 is

$$\underline{r}_O = G_{LV}^T (\underline{R}_T - \underline{R}_S)$$

Dividing each side of 7.1.1 by the initial range

$$\dot{\underline{r}}_O / r_O = F \underline{r}_O / r_O$$

$$= F \underline{X}_{LOS} \quad 7.1.2$$

by definition. Now convert to LOS coordinates

$$G_{LOS}^T \dot{\underline{r}}_O / r_O = [v_{XLOS}/r_O, v_{YLOS}/r_O, v_{ZLOS}/r_O]$$

$$= [\dot{r}/r)_O, \dot{\eta}_O, \dot{\alpha}_O]$$

Where $\dot{\eta}$ and $\dot{\alpha}$ are the required normal to the line-of-sight angular rates, and $(\dot{r}/r)_O$ is required ratio of range rate to range. Direct readings of r_s , $\dot{r}_s$, $\dot{\eta}_s$, $\dot{\alpha}_s$ are now obtained from the rendezvous sensor, from which $\underline{r}_O$ and the required rates computed. The delta-v required is then

$$v_{XLOS} = \dot{r}_s - (\dot{r}/r)_O r_s$$

$$v_{YLOS} = - (\dot{\alpha}_s - \dot{\alpha}_O) r_s$$

$$v_{ZLOS} = - (\dot{\eta}_s - \dot{\eta}_O) r_s$$